REPORT DOCUMENTATION PAGE

AFRL-SR-BL-TR-01-

0121

1. AGENCY USE ONLY (Leave blank)	2. REPORT DATE	3. REPORT TYPE A[ND DATES] COVERED
	September 2000	Final Report 1/1/99--6/30/00

4. TITLE AND SUBTITLE	5. FUNDING NUMBERS
Simulation of Structural Bending modes of Large Aircraft	G5368 Contract #F49620-99-1-0093
6. AUTHOR(S) Dr. Daniel Biezad	
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) California Polytechnic State University San Luis Obispo, CA 93407	8. PERFORMING ORGANIZATION REPORT NUMBER
9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES) Air Force Office of Scientific Research Edwards AFB CA	10. SPONSORING / MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION AVAILABILITY STATEMENT

Approved for public release, distribution unlimited

12b. DISTRIBUTION CODE

13. ABSTRACT (Maximum 200 Words)

Large aircraft may possess slow structural modes that can affect handling qualities if excited. Little has been done to simulate these structural modes during pilot-in-the-loop analyses. A fast and simple method to simulate the longitudinal structural modes of large aircraft has been developed. A shape function for fuselage was obtained from a finite element model. This deflection function was used to develop transfer functions for flight simulation. The transfer functions were developed in a method similar to that used at NASA Dryden Flight Research Center. Flight simulations were developed to explore the effects of the structural modes on handling qualities of the aircraft in question. These calculations were validated by correlating wind tunnel data with simulation predicted data.

14. SUBJECT TERMS	15. NUMBER OF PAGES
Structural Dynamics, Simulation	16
	16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std Z39-18
298-102

INVESTIGATION OF DYNAMIC STRUCTURAL MODELS SUITABLE FOR THE SIMULATION OF LARGE AIRCRAFT

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June 2000

ABSTRACT

SIMULATION OF STRUCTURALBENDING MODES OF LARGE AIRCRAFT

By Aaron R. Munger

Large aircraft may possess slow structural modes that can affect handling qualities if excited. Little has been done to simulate these structural modes during pilot-in-the-loop analyses. A fast and simple method to simulate the longitudinal structural modes of large aircraft has been developed. A shape function for fuselage was obtained from a finite element model. This deflection function was used to develop transfer functions for flight simulation. The transfer functions were developed in a method similar to that used at NASA Dryden Flight Research Center. Flight simulations were developed to explore the effects of the structural modes on handling qualities of the aircraft in question. These calculations were validated by correlating wind tunnel data with simulation predicted data.

ACKNOWLEDGEMENTS

Acknowledgement and appreciation are owed to Mr. Fred Webster and Mr. Kurt Buehler of the Air Force Flight Test Center, Edwards Air Force Base, and to Dr. Len Sakell of the U.S. Air Force Office of Scientific Research, for their support of this research project. The support and guidance of Dr. Daniel Biezad, Dr. Faysal Kolkailah, and Dr. Eltahry Elghandour were important in bringing this project to a successful completion. The interaction with and support of Wendy Hashii were also instrumental during the course of the project. Special thanks to Jonathan Almond for his expertise and support, as well as to committee members Dr. Don Hartig and Dr. Roger Ludin. Finally, thanks to all of those who supported and assisted in making this work possible.

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NOMENCLATURE

a_i	polynomial coefficients
A	system matrix
A_1	mode shape bias
A_2	mode shape angular scaling factor
A_{n_s}	normal acceleration due to structural bending
A(s)	actuator transfer function
B	control matrix
c	viscous damping coefficient, lb-sec/ft
cg	center of gravity
C	output matrix
D	output control matrix
E	modulus of elasticity, psi
F	harmonically varying force, lbf
F_1	maximum value of force, lbf
F_s	stick force, lbf
F_δ	control surface input effectiveness, in/deg
FE	finite element
FS	fuselage station, in
FS_0	reference fuselage station, in
g	gravitational constant, ft/sec^2
GVT	ground vibration test
H(s)	Laplace transfer function, feedback loop
I	identity matrix
I_x	moment of inertia about x axis, similar for y and z
I_{xy}	product of inertia about x and y axes, similar for other axes
K_1	displacement constant
K_θ	pitch attitude gain, deg/deg

L	characteristic length, in
LCO	limited cycle oscillation
p	roll rate
PIO	pilot-induced oscillation
q	pitch rate
r	yaw rate
RPO	residual pitch oscillation
s	Laplacian operator
SAS	stability augmentation system
t	time, sec
$\mathbf{u}$	control vector
u	forward air speed
v	horizontal air speed
w	vertical air speed
$\mathbf{x}$	state vector
$\dot{\mathbf{x}}$	derivative of state vector
x	horizontal axis coordinate, in
$\dot{x}$	velocity, in/sec
$\ddot{x}$	acceleration, in/sec^2
x_{ss}	steady-state solution, in
$\mathbf{y}$	output vector
y	horizontal position
z	vertical position
δ	deflection
δ_e	elevator deflection, deg or rad
δ_{e_s}	elevator deflection due to stick position, rad
$\Delta\delta$	elevator deflection from trim
ζ	damping ratio
η	structural modal displacement, in

$\dot{\eta}$	structural modal velocity, in/sec
$\ddot{\eta}$	structural modal acceleration, in/sec^2
θ	pitch angle
$\dot{\theta}$	pitch rate
ϕ	nondimensional constant
Φ	roll angle
ψ	heading angle
ω	frequency, rad/sec
ω_d	damped natural frequency, rad/sec
ω_n	natural frequency, rad/sec

Subscripts

0	bias term
d	damped natural frequency
e	elevator
n	natural frequency
s	structural value
ss	steady state value

CHAPTER 1

INTRODUCTION

Background

The strength and flexibility characteristics of large, modern aircraft structures often produce structural modes of vibration that are of the same order of magnitude as the bare airframe short-period response. The first bending mode of the structure may in this case have an effect on the handling qualities of the aircraft and should considered in a piloted simulation of the vehicle. Currently, piloted simulations do not include structural modes, most of which are highly dependent on configuration. MIL-STD 1797 Flying Qualities of Piloted Aircraft[1] does not provide metrics for the handling quality related structural modes of an aircraft.

Variable Definitions

In this investigation, a large aircraft (see Figure 1) is considered a combination of a fuselage (including the tail) and right and left wings. The wings and tail provide excitation inputs to the flexible fuselage, which is allowed to bend as viewed from the side but not allowed any degrees of freedom in twist.

Figure 1. Large Aircraft in Flight.

The aircraft is free to pitch about its center of gravity, and the center of gravity is allowed to shift slightly along the longitudinal axis. When the airplane is disturbed

from its equilibrium state, the resulting motion in the longitudinal plane may be considered the sum of the motion due to the nonlinear equations of motion plus the linear vibration oscillation.

When considering the bending modes due to aeroelastic effects of large aircraft a few system parameters must be defined in order to provide a clear understanding of the problem. At any instant during flight a flexible aircraft fuselage can take on the shape similar the one shown in Figure 2. The dark blue line represents the deformed shape of

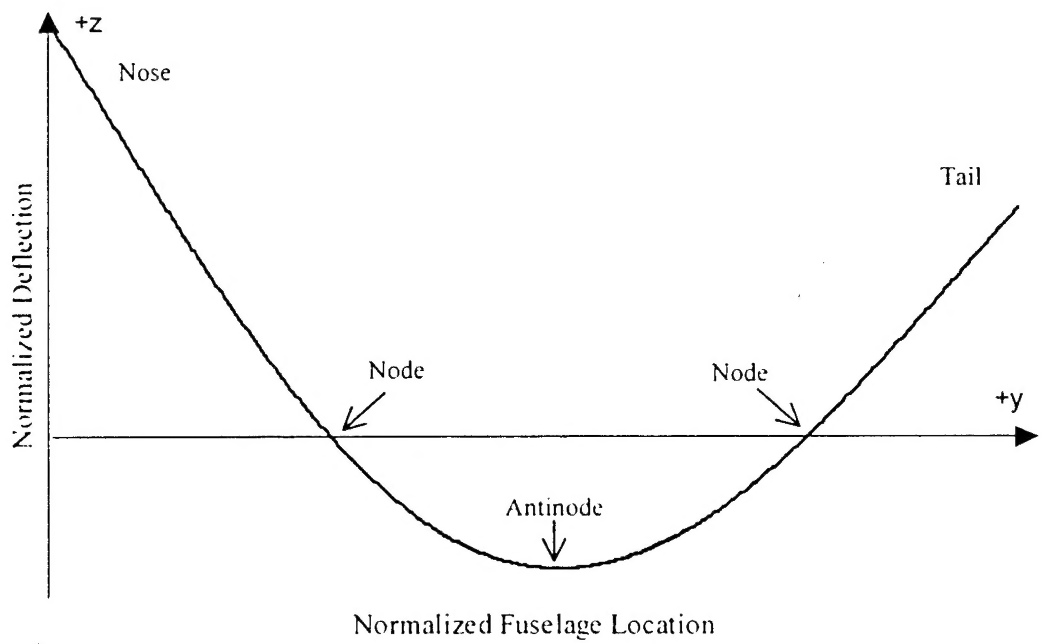

Figure 2. Coordinate Systems for a Deflected Aircraft Fuselage

the fuselage during flight. The mode shape is defined as the deformed shape which the fuselage of a flexible aircraft takes on during flight. The horizontal axis is the normalized fuselage position. The normalized fuselage position is used to determine the placement of items such as accelerometers and gyros along the longitudinal axis of the aircraft. Normalized fuselage position is obtained by taking the horizontal reference system and dividing all values by the overall fuselage length. Normalized deflection of the fuselage is shown on the vertical axis of Figure 2. Normalized deflection is defined as the amount which the fuselage deflects from a given reference system. The deflections are normalized by determining the maximum deflection (usually at the nose of the aircraft) and then dividing the deflection distribution by this parameter. Figure 2 also shows the two nodes of the first bending mode. A node is a point of zero displacement and is represented by the points where the deflected fuselage (the dark blue line) crosses the horizontal axis. Nodes are points along the fuselage which experience no transnational motion only rotational motion. An aircraft can have any number of nodes depending on the mode of vibration which is being excited. The first bending mode of the aircraft is shown here in Figure 2. The first bending mode is observed when the mode shape of the fuselage has two distinct nodes as shown in Figure 2. The second bending mode is observed when three distinct nodes are present along the fuselage. The node and bending mode relationship can continue to infinity governed by the following equation:

$$\text{Bending Mode} = \text{Total Number of Nodes} - 1$$